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AN EXPERIMENTAL INVESTIGATION OF HEAT TRANSFER IN THE NOZZLE OF HIGH-ALUMINIZED SOLID ROCKET (EFFECTS OF ALUMINUM ADDED TO PROPELLANT)

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ABSTRACT. The heat transfer in the nozzle of a high-aluminized solid rocket has been experimentally investigated.

It is noted that the addition of metallic fuel to the propellant has a pronounced effect on the heat transfer coefficient but that the radiant heat flux is not dominant in such a nozzle.

It is also notec that Bartz's treatment may be applied to the heat transfer problem in a mixed gas-particle flow, assuming that the gas-particle flow behaves like the gas flow with equivalent specific heat ratio $\overline{\kappa}$.

The equivalent specific heat ratio is obtained, where the velocity-lag of the particle behind the gas flow is negligibly small.

An experimental study was made of the heat transfer in a solid rocket mortar nozzle using a polybutadiene system composite propellant containing aluminum. It was found that the rate of heat transfer increases markedly as the aluminum content in the propellant is increased, and that the effect of heat radiation of alumina particles was small. Moreover, in computing the heat transfer, the combustion gas flow containing alumina particles was treated as a single gas phase flow having mixed phases and equivalent specific heats, and it was found that the results of computation obtained by applying the semi-theoretical equation pertaining to turbulent flow heat transfer within a tube matches well with the experimental values.

1. INTRODUCTION:

Characteristics such as the specific thrust of solid rocket propellants have improved greatly in recent years. There are reasons for this progress; we can consider advances in research on oxidants, fuel binders, and catalysts which are propellant components, together with the effects obtained by the addition of metal fuels such as aluminum powder to them. However, while the addition of metal fuels serves to improve the specific thrust, they make the heat conditions in the nozzle section more severe. In propellants containing large quantities of aluminum, there is much greater damage done to the nozzle throat section in comparison with propellants not containing aluminum, and ablation is particularly severe with plastic nozzles. Because of this it is considered to be the elevation of temperature of the combustion /1*

*Numbers in the margin indicate pagination in the foreign text.

gas due to the burning of the aluminum, and the effects of the coagulated particles of alumina. It is a well known fact that if one assumes that the combustion gas is a perfect gas, and that the reaction in the process of expansion in the nozzle is frozen, the results of computation obtained by application of Bartz's semi-theoretical equation [1,2] for turbulent flow heat transfer within a tube agree well with experimental values, in the case of a heat transfer within a rocket motor nozzle using propellants that do not contain metal additives. However, the weight ratio of alumina in the combustion products in a propellant containing 20% aluminum, reaches as much as 50%, and the method of treating the effects of this ratio poses a problem.

We have conducted an experimental study of heat transfer in a solid rocket motor nozzle, using a polybutadiene system composite propellant, containing no aluminum, 10% aluminum and 20% aluminum, and have attempted to do a theoretical analysis of the results. /2

2. SYMBOLS:

- A: Area
- C_p: Specific heat at constant pressure
- D: Nozzle diameter
- E: Nozzle throat area ratio
- F: Thrust
- I: Volume ratio
- I_{sp}: Specific Thrust
- M: Mach
- P: Combustion pressure
- P_r: Prandtl number
- Q: Heat capacity
- R: Gas constant
- W: Weight
- a: Average cross section area of particle
- c: Specific heat
- c*: Characteristic exhaust gas rate
- d: Particle diameter
- f: Configuration coefficient
- g: Acceleration of gravity
- h: Heat transfer coefficient
- k: Heat conductivity
- l: Distance
- m: Average molecular weight of gas
- n: Number of particles per unit volume
- q: Heat transfer volume
- r: Radius of curvature of nozzle throat
- t: Time
- u: Velocity
- γ: Specific Weight

ε: Monochromatic radiation rate
θ: Temperature
μ: Viscosity coefficient
ρ: Density
σ: Boltzmann's constant
κ: Specific heat ratio

SUBSCRIPTS:

c: Convection flows
g: Gas
o: Stagnation point
p: Particle
pp: Propellant
r: Radiation
t: Throat
w: Wall
x: Point

3. METHOD OF EXPERIMENT

In this investigation, it was decided to determine the rate of heat transfer by measuring the temperature changes of plugs placed perpendicular to the wall surface in several places within the nozzle. Many thermocouples are required in this method, but it is well known that in cases such as a solid rocket, the combustion time is relatively short, and it is possible to obtain highly acurate resul:s in the case where it is difficult to obtain a thermally stable state [3].

If we consider here that the surrounding surface of the plug which is used for measuring heat transfer is completely heat-insulated except on the heat transfer surface and in the e.tremity, it is possible to view the plug, in terms of heat, as a single dimension model. The increase in heat capacity per unit of time can be expressed as

$$\dot{q}=\frac{1}{A}\frac{\partial}{\partial t}\int_0^l AQ\,dx+k\left(\frac{\partial\theta}{\partial x}\right)_l \qquad (1)$$

since it is equal to the heat flux $\dot{q}$ which enters through the heated surface, and if we assume that the heat loss from the outer end is $k\left(\frac{\partial\theta}{\partial x}\right)_l = 0$, then the rate of heat flow into a plug with a length, l, and a constant cross section area, A, in the time interval between t_1 and t_2 is:

$$\dot{q}_{(t_2\cdot t_1)}=\frac{1}{t_2-t_1}\left[\int_0^l(\rho c\theta)_{t_2}d_x-\int_0^l(\rho c\theta)_{t_1}d_x\right] \qquad (2)$$

Consequently, the heat transfer rate h is computed by

$$h_{\frac{1}{2}(t_1+t_2)}=\frac{1}{(\theta_g-\theta_w)_{\frac{1}{2}(t_1+t_2)}}\times\left\{\frac{1}{t_2-t_1}\left(\int_0^l(\rho c\theta)_{t_2}d_x-\int_0^l(\rho c\theta)_{t_1}d_x\right)\right\} \tag{3}$$

The plugs were made of pure copper which have a high temperature conductivity, in order to avoid losses on the heating surfaces, and in order to minimize the heat exchange with the surrounding parts, the plug was placed in a nozzle made of the same material. /3

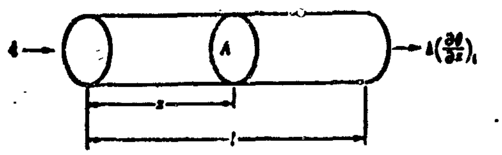
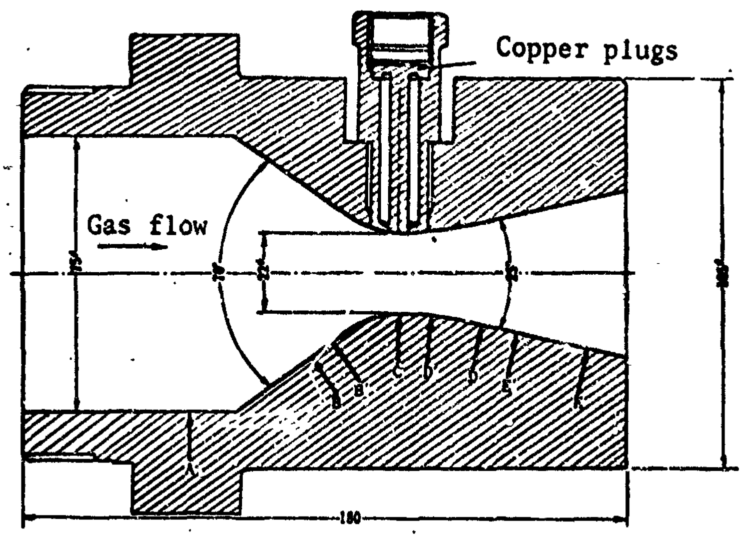


Fig. 1: Model of Copper Plug

As shown in Figure 2, two models (I, II) with differing plug positions for temperature measurement were used for the nozzles that were tested. In model (I) the five points A, B, C, D, E, and in model (II) the five points A, B', C, D'. E' are the positions of plug attachment. In each case C corresponds to the throat section.



Position / Nozzle type	Plug position and throat area ratio							
	A	B	B'	C	D'	D	E'	E
I	11.6	4.5		1.0		2.0		4.0
II	11.6		3.0	1.0	1.3		2.6	

Fig. 2: Cross Section and Plug Posit'on in Nozzle Used for Measuring Heat Transfer

As indicated in Figure 3, two holes one mm deep and 0.4 mm in diameter were made at a distance of 1 mm from each other on the periphery of one plug. After inserting an alumel wire 0.3 mm in diameter and a chromel wire of the same diameter into these holes, thermocouples were placed in four spots at distances of 8, 13, 20, and 28 mm from the end of the heated area. A total of 20 thermocouples was used. In order to reduce the area of contact of the plug and the nozzle wall, they were made to fit closely and the section of the plug protruding into the nozzle inner wall was shaved off, so that it was uniform with the nozzle inner wall. This means that the distance from the heated surface to the thermocouple differed from the 8, 13, 20, and 28 mm described above, so after tests, these were measured acurately by removing the plugs. Furthermore, in order to reduce the heat fluctuation due to conduction currents in the space around the plug, the gap between it and the nozzle itself was reduced.

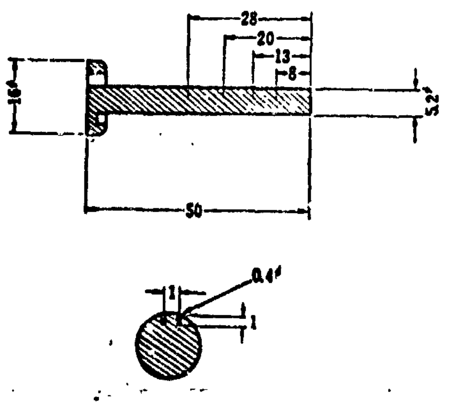


Fig. 3: Plug Configuration and Position of Thermocouple Insertion

The propellant used in the tests was a polybutadiene composite with ammonium perchlorate as the oxidant. The configuration is shown in Figure 4. The diameter is 93 mm, length 327 mm and weight 2.6 kg. Since the heat transfer rate differs depending on the combustion gas pressure, flow velocity

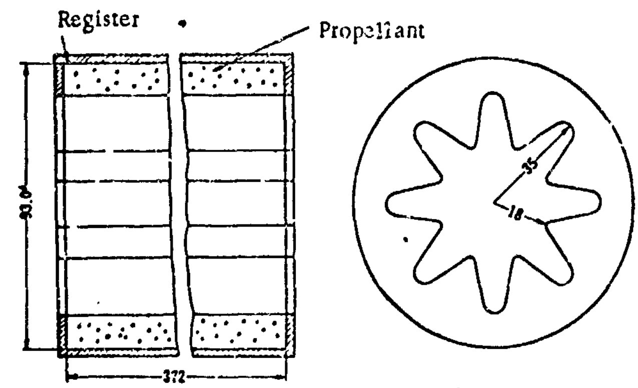


Fig. 4: Configuration of Propellant

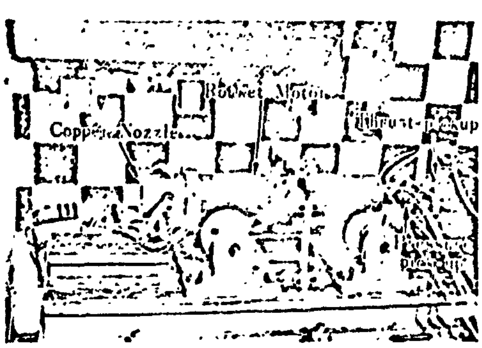


Fig. 5.

and flow rate, the internal configuration of the propellant was selected so that these values would remain as constant as possible during measurement of heat transfer, i.e. so that the thrust would remain as constant as possible. For the propellant three varieties were used, one not containing aluminum powder, one with 10% and one with 20% aluminum. The average particle diameter of the aluminum powder used was 19 μ. The propellant composition is shown in Table 1. /4

The rocket motor burning test was conducted on a one ton lateral test stand in the solid rocket laboratory. Figure 5 shows the rocket motor on the test stand. Resistance wire strain gauge type pick-ups were used for measuring the thrusts and combustion pressure, and the apparatus was designed so that these measurements, along with measurements of temperature could be recorded on a direct vision electromagnetic oscillograph. The thermocouples used for measuring temperature throughout the measuring system, were calibrated at 20°C, 60°C and 80°C, with the melting points of tin, lead, and zinc prior to the tests.

Table 1.: Composition of Propellant

Composition / Propellant	Ammonium perchlorate	Polybutadiene	Aluminum Powder
Propellant with 0%* Al	80	20	0
Propellant with 10%* Al	72.7	18.2	9.1
Propellant with 20%* Al	66.6	16.7	16.7

*Percentage increase to gross weight

The burning tests for each propellant were conducted under conditions where K_N (ratio of combustion area to nozzle throat area) were constant. As explained below, for propellants with a 0% and 10% aluminum content data were obtained for four burning tests, but in the case of propellant containing 20% aluminum, in the two burning tests that were conducted, the nozzle throat was burned up 0.6 seconds after ignition both times. For this reason, for the propellant with 20% aluminum, the pure copper nozzle was replaced with a graphite nozzle and separate burning tests were conducted.

4. Test Results

The results of burning tests of each propellant are shown in Table 2 below.

Table 2.: Burning Tests Results

Type of Propellant		Propellant with 0% Al				Propellant with 10% Al				Propellant with 20% Al		
Observation \ Test		1	2	3	4	1	2	3	4	1	2	3*
Minimum Pressure	P_{min}	39.3 kg/cm²	38.5	41.5	41.2	48.5	49.5	53.3	52.5	46.9	53.7	48.5
Maximum Pressure	P_{max}	52.3 kg/cm²	54.4	55.1	59.5	69.5	70.3	71.1	70.3	—	—	60.2
Propellant Weight	W_{pp}	2.486 kg	2.478	2.477	2.468	2.590	2.585	2.576	2.545	2.676	2.683	2.677
Specific Thrust	I_{sp}	213 sec	—	211	211	232	221	226	227	—	—	226

* Results from graphite nozzle

One example of the thrust-time curve is shown in Figure 6. In each case /5
the thrust was 300 kg, the combustion pressure 50 kg/cm^2, and the burning time was two seconds. It was decided to read the temperature data for calculating the heat transfer rate off the recording paper in the interval during which the burning state can be regarded as almost steady, avoiding the unsteady burning time immediately after ignition, since the heat transfer rate is largely determined by the physical values of the gas. In Figure 7, the distance from the thermocouple on the copper plug to the nozzle wall is parted on the x-axis and the temperature at these points is parted on the y-axis (logarithmic axis). The wall temperature θ_w is determined by extrapolation since it is expressed by a primary equation in a theoretically semi-logarithmic coordinate system, if one assumes that there is no loss from the periphery of the nozzle, and the heat transfer rate is also steady. Moreover, the temperature distribution between each measuring point within the plug is expressed as a primary equation, and the heat transfer volume $\dot{q}_{1/2}(t_1 + t_2)$ was calculated from equation (2).

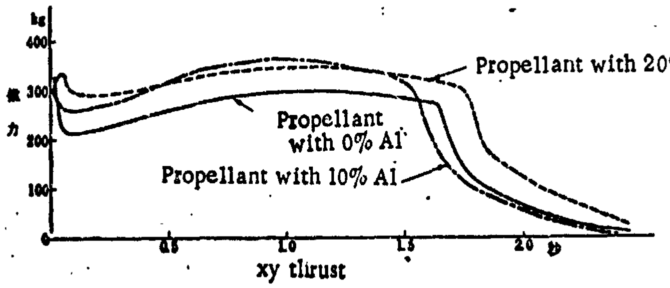


Fig. 6: Thrust-time Curves

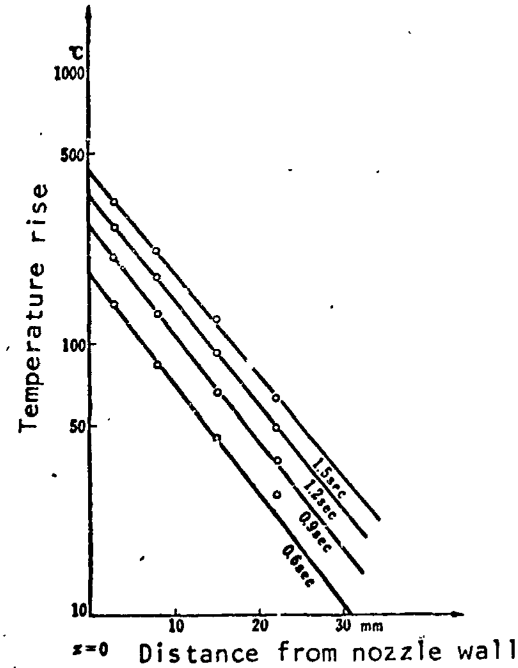


Fig. 7: Temperature Distribution on Plug at Point E' in the Case of a Propellant with 10% Al

5. ANALYSIS

5.1 Alumina particles in combustion gas.

It is considered that the diameter and size of alumina particles produced by the burning of aluminum gradually increase as the alumina themselves collide and fuse together repeatedly within the nozzle [4, 5, 6]. However, there are reports concerning the particle diameter of alumina which is ejected from the nozzle, stating that the diameter varies according to combustion pressure [7], and reports that the diameter is 2-3 μ regardless of combustion pressure [8]. However, it is considered a fact that the diameter of the alumina particles varies according to the aluminum content in the propellant [9, 10].

Even if the diameter of the alumina particles in the nozzle increases due to collision and bonding, in the case of propellant containing 20% aluminum, it is reported that the diameter of particles ejected through the nozzle is below 4.5 μ at a combustion pressure of 70 kg/cm² [10], and even if one calculates and compares the gas velocity and the alumina particle velocity inside the nozzle, there is no great difference when the particle diameter is less than several μ, and the temperature difference between the gas and the alumina particles is small.

Here we shall consider the flow in the nozzle on the basis of the following conditions.

(1) There is no loss of mass or energy from the system.

(2) There is no variation in mass between phases.

(3) The alumina particles do not cause a reaction, and the volume occupied by them may be disregarded.

(4) The gas is a constant component and is considered to be a perfect gas.

(5) The specific heat of the gas and alumina particles is assumed to be constant.

The energy equation for the gas and alumina particles is

$$\dot{W}_g\left[c_{pg}(\theta_g-\theta_{g0})+\frac{1}{2}u_g^2\right]$$

$$+\dot{W}_p\left[c_{pp}(\theta_p-\theta_{p0})+\frac{1}{2}u_p^2\right]=0 \qquad (4)$$ /6

As we have already noted, since the alumina particles that are produced are very small, if we assume that the temperature difference and the velocity difference between the gas and the particles may be disregarded, we can find the equivalent specific heat ratio of the mixed flow of particle and gas $\bar{\kappa}$. If, in equation (4) we assume $u_p = u_g = u$, $\theta_p = \theta_g = \theta$

$$\bar{\kappa}=\kappa\left[\frac{1+\left(\frac{W_p}{W_g}\right)\left(\frac{c_{pp}}{c_{pg}}\right)}{1+\kappa\left(\frac{W_p}{W_g}\right)\left(\frac{c_{pp}}{c_{pg}}\right)}\right] \qquad (5)$$

Here, κ is the specific heat ratio only for the gas phase.

That is, we can regard the two phase flow of gas and particle as a gas phase in which the specific heat ratio is $\bar{\kappa}$.

If we consider the nozzle gas to be an ideal gas, the gas velocity in the nozzle u_g is

$$E=\frac{A_x}{A_t}=\frac{M_t}{M_x}\sqrt{\left[\frac{1+\frac{\kappa-1}{2}M_x^2}{1+\frac{\kappa-1}{2}M_t^2}\right]^{\frac{\kappa+1}{\kappa-1}}} \tag{6}$$

If the specific heat ratio κ is determined, it is possible to compute the Mach number M_x at point x with respect to the aperture ratio E.

5.2 Physical values of combustion products

Since in this test we are not measuring the combustion gas temperature, specific heat ratio, or the average modular weight, these values were determined from the results of tests on the composition of the propellant, combustion pressure and specific thrusts, and from the literature [9, 10, 11].

The specific heat ratio is calculated from equation (5) but the percentage of alumina contained in the products of combustion in equation (5) W_p/W_g was determined assuming that all of the aluminum underwent reaction in the propellant, forming alumina. The results are shown below in Table 3.

Table 3: Alumina Weight Ratio in Products of Combustion

	$\frac{W_p}{W_p+W_g}$
Propellant with 0% Al	0
Propellant with 10% Al	0.17
Propellant with 20% Al	0.48

5.3 Convection flow heat transfer rate

Considerable tests of heat transfer in nozzles of rocket motors, which do not contain powdered metal such as aluminum, have been made up to the present time, and semi-theoretical equations have been obtained. Of these, Bartz's equations agreed well with the test values, and are used in calculating heat transfer in nozzles. The convection flow heat transfer volume $\dot{q}_c$

$$\dot{q}_c=h_c(\theta_g-\theta_w) \tag{7}$$

Here, using Bartz's semi-theoretical equation, the convection heat transfer rate h_c is

$$h_c=\left[\frac{0.026}{D_t^{0.2}}\left(\frac{\mu^{0.2}c_p}{P_r^{0.6}}\right)\left(\frac{pg}{c^*}\right)^{0.8}\left(\frac{D_t}{r}\right)^{0.1}\left(\frac{1}{E}\right)^{0.9}\right]\delta \tag{8}$$

In this equation the heat transfer volume is validly determined by the percentage of flow of mass per unit area.

$$\delta=\frac{1}{\left[\frac{1}{2}\frac{\theta_w}{\theta_0}\left(1+\frac{k-1}{1}M^2\right)+\frac{1}{2}\right]^{0.63}\left[1+\frac{k-1}{2}M^2\right]^{0.13}} \tag{9}$$

The convection flow heat transfer was computed on the basis of (7).

5.4 Radiant Heat Transfer

The radiant heat transfer volume $\dot{q}_r$ from the alumina particle cloud to the nozzle wall is expressed by the following equation.

$$\dot{q}_r=\varepsilon_w\cdot\varepsilon_p\cdot\sigma\cdot f(\theta_p{}^4-\theta_w{}^4) \tag{10}$$

As for the radiant heat transfer between particles, we consider the alumina particle to be spherical, and assuming that the temperature of a particle is almost equivalent to that of the neighboring particle, and that the heat transfer within the ambient gas is negatibly small, if we assume that heat transfer occurs only from the alumina particles to the nozzle wall, f = 1, and equation (10) becomes

$$\dot{q}_r=\varepsilon_w\cdot\varepsilon_p\cdot\sigma(\theta_p{}^4-\theta_w{}^4) \tag{11}$$

or

$$\varepsilon_p=1-e^{-nal} \tag{12}$$

(See 12).

Expressing the gas velocity at a point where the nozzle inner diameter is D as u_g, and the specific weight of the alumina particles as γ_p, since the gas volume passing through the nozzle in a unit of time is $\frac{\pi}{4}\cdot D^2\cdot u_g$, the relative volume of the generated gas and the alumina particles I is

$$I=\frac{\frac{\dot{W}_p}{\gamma_p}}{\frac{\pi D^2 u_g}{4}} \tag{13}$$

$\dot{W}_p$ is the alumina particle volume of particles passing through the nozzle in a unit of time.

$$\dot{W}_p=\frac{W_p}{W_p+W_g}\cdot\frac{F}{I_{sp}} \tag{14}$$

Whereupon,

$$nal=\left(\frac{l}{\frac{\pi}{6}d_p^3}\right)\cdot\frac{\pi}{4}\cdot d_p^2\cdot\frac{D}{2}=\frac{3lD}{4d_p} \tag{15}$$

Substituting equations (13) and (14) in equation (15),

$$nal=\frac{3}{\pi D\cdot u_g\cdot\gamma_p\cdot d_p}\left(\frac{W_p}{W_p+W_g}\right)\cdot\frac{F}{I_{sp}} \tag{16}$$

Therefore, the effective radiation rate ε_p of the alumina particle cloud can be determined, given small use of u_g, γ_p, and d_p. Moreover, the radiation rate of the nozzle wall ε_w can be considered to be almost constant when the nozzle wall temperature is below 1000°C. From the above, the radiant heat transfer rate $\dot{q}_r$ can be computed by equation (11).

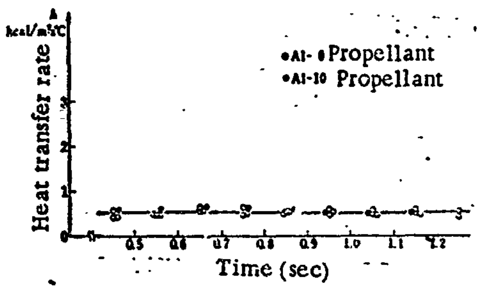


Fig. 8: Variation in Heat Transfer Rate at Point A

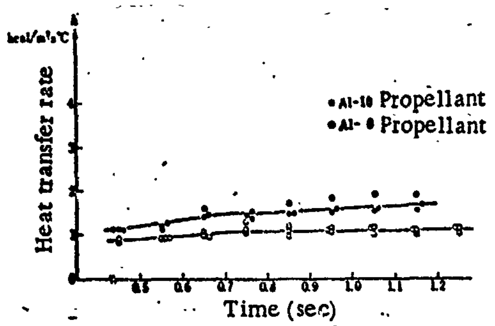


Fig. 9: Variation in Heat Transfer at Point E'

6. Examination of Results /9

Figures 8 and 9 show variation in the heat transfer rate at points A and E' during combustion. This variation is almost stable with respect to the passage of time. This is because the flow rate of products of combustion passing through the nozzle during that interval is almost constant and as shown in Figure 10, if we determine the heat transfer rates at 0.5-0.6 seconds and 1.0-1.1 seconds and compare them, we see that they are almost constant regardless of the presence or absence of aluminum in the propellant.

Figure 11 indicates the results of tests on the effect of the percentage of aluminum contained in the propellant on the heat transfer rate. Clearly, as the amount of aluminum is increased, the heat transfer rate shows larger values, in particular a value two times greater is indicated in the throat section. As for the heat transfer rate in the case of propellant containing no aluminum, 0.5-0.6 seconds after ignition, as shown in Figure 12, the experimental values and the theoretical values for the convection flow heat transfer rate agree well, except in the fan-shaped section of the nozzle. There is a good match between experimental and theoretical values 1.0-1.1 seconds after ignition, as shown in Figure 15, except in the throat section. Figures 13 and 14 show the distribution of the heat transfer rate in the case of propellant containing 10% and

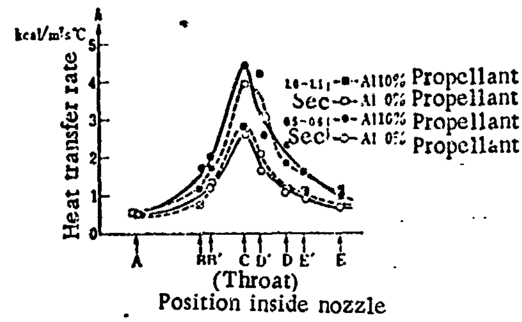


Fig. 10: Comparison of Heat Transfer Rate in Nozzle at Time 0.5-0.6 Seconds and 1.0-1.1 Seconds

20% aluminum respectively, at 0.5-0.6 seconds. It can be seen here, that there is a rather good match between computed values of conduction flow heat transfer and test values. The magnitude of the radiant heat transfer from the aluminum particles is smaller than the conduction flow heat transfer, but the computed values in which corrections are made for radiation, indicate a closer approximation with respect to the test values. The radiant heat transfer rate is particularly great in the vicinity of the nozzle inlet. Figures 15 and 16 indicate the distribution of the heat transfer rate 1.0-1.1 seconds after ignition. Generally a slight difference can be observed between theoretical and test values in the nozzle throat section, but this is considered to be due to scattering in the tests.

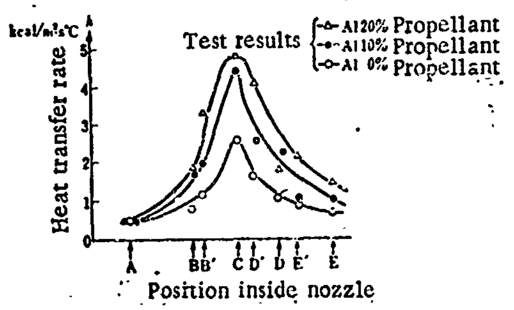


Fig. 11: Effect of Aluminum Content in Propellant on the Heat Transfer Rate in the Nozzle Section (Test Results 0.5-0.6 Seconds after Ignition)

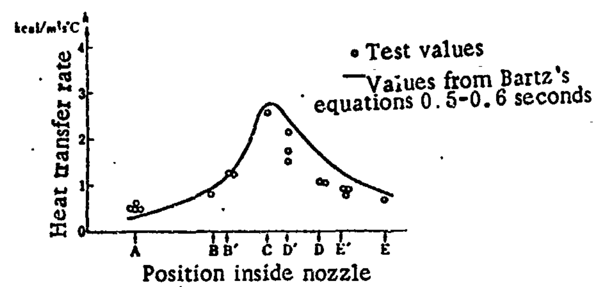


Fig. 12: Heat Transfer Rate Inside Nozzle in the Case of Propellant with 0% Al

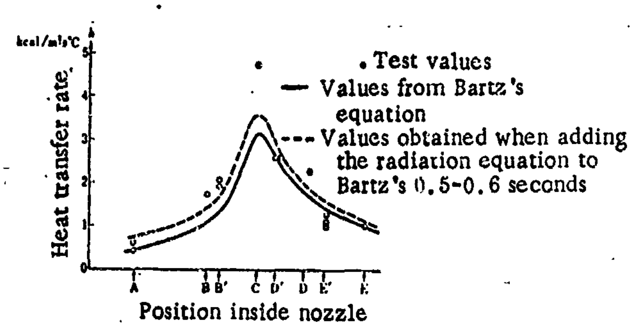


Fig. 13: Heat Transfer Rate Inside Nozzle in Case of Propellant with 10% Al

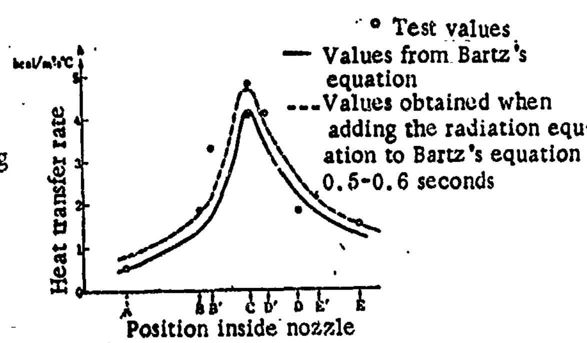


Fig. 14: Heat Transfer Rate Inside Nozzle in the Case of Propellant with 20% Al

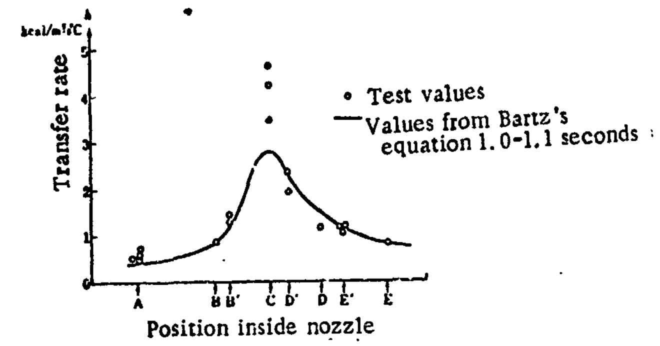


Fig. 15: Heat Transfer Rate Inside the Nozzle in the Case of Propellant with 0% Al

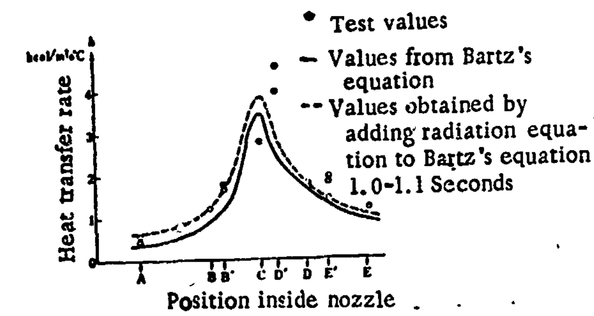


Fig. 16: Heat Transfer Rate Inside the Nozzle in the Case of Propellant with 10% Al

7. Conclusion

The following facts were determined from this experimental study.

(1) There is a marked increase in the heat transfer rate inside the nozzle as the amount of aluminum contained in the propellant is increased.

(2) Even when the propellant contains 20% aluminum, the influence of radiation from aluminum particles on the nozzle heat transfer is small, the major influence being from conduction flow heat transfer.

(3) The computed values for the conduction flow heat transfer almost match the test values when computed using Bartz's semi-theoretical equation, even for propellant containing large amounts of aluminum, if we assume the mixed flow of gas and alumina particles go through the nozzle as a gas phase with an equivalent specific heat ratio $\bar{\kappa}$.

Finally, we would like to express our gratitude to the personnel in the rocket division beginning with Division Chief Kuroda, and to Technical Officer Sekine of the Engine Section for their guidance and cooperation throughout this study.

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